

NUMERICAL AND EXPERIMENTAL SIMULATION OF MEDIUM COMPRESSION EFFECT AT FLOWING AROUND A WING PROFILE

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Introduction

Simplified theoretical models are quite often used in aerodynamical investigations to solve the problems of flowing around the aircraft elements with a compressed flow, in particular, the model of incompressible liquid based on using of the known correction factor which is a function of the Mach number. The similar approaches have a whole series of restrictions with conditional and interdependent limits. By this, a trusty information made in the frames of models verification, such as a pressure distribution on the surface, or nicer characteristics, for example, flow downwards, even by flowing around bodies of a non-complicated form is not exhaustive by no means. This work is to make up a deficiency, in which on the example of flowing in the vicinity of a symmetrical wing profile with a relative thickness of 12 % the numerical and experimental data have been obtained. These data allow to evaluate an adequacy level of different modeling ways of the compressed medium effects.

Experimental conditions and methods

The first part of the experiment was conducted in sub-sonic low-turbulent wind tunnel T-324 of ITAM SB RAS. The investigated model represents a rectangular symmetrical wing including profile sections of NACA0012 type. It was drained by static pressure tubes with diameter of 0,5 mm posed non-uniformly in a central principal plane along distant from an observer and partially along a nearer profile element. To prevent possible end effects and influence of the tube walls boundary layer on the flow characteristics in the investigation working field, and also to increase an effective wing lengthening, the model is provided with end aerodynamic washers. The prior experiments show that flowing along such wing is performed by a plane stream and in essence, is analogous to a regime of flowing along the endless lengthening wing.

The experiments were conducted at free-stream velocities $U_\infty = 10, 25, 35, 50$ m/s that corresponded to the Reynolds' numbers along the wing chord $Re_c = 3.34 \cdot 10^5, 8.33 \cdot 10^5, 1.17 \cdot 10^6, 1.66 \cdot 10^6$. A flow velocity was controlled on a differential between the pressure drag in the tube pre-chamber and a static pressure in the test section, and also was duplicated by measurement of the full and static pressure using the Pitot – Prandtl tube posed in the controlled section. It was not applied an artificial turbulization of the boundary layer to provide a laminar flow form on the wing profile surface wherever possible, and as a consequence, to decrease the viscous effects on the flow compression. It was found during the experiments that position of the boundary layer transition from laminar to turbulent is smoothly shifted in direction of the leading wing edge as the free-stream velocity is increased.

The second part of the experiments was conducted at the transonic wind tunnel T-205M FGUP SibRAC named by S.A. Chaplygin. The investigated model is the same. Special methodical experiments showed that flowing is created by practically plane stream and is essentially analogous to the flowing regime along endless lengthening wing. The experiments were performed at the Mach numbers of the free-stream $M_\infty = 0.4; 0.5; 0.6; 0.65; 0.7; 0.73; 0.81$,

Report Documentation Page

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|---|--|--|
| Report Date 23 Aug 2002 | Report Type N/A | Dates Covered (from... to) - |
| Title and Subtitle Numerical and Experimental Simulation of Medium Compression Effect at Flowing Around A Wing Profile | | Contract Number |
| | | Grant Number |
| | | Program Element Number |
| Author(s) | | Project Number |
| | | Task Number |
| | | Work Unit Number |
| Performing Organization Name(s) and Address(es) Institute of Theoretical and Applied Mechanics Institutskaya 4/1 Novosibirsk 530090 Russia | | Performing Organization Report Number |
| Sponsoring/Monitoring Agency Name(s) and Address(es) EOARD PSC 802 Box 14 FPO 09499-0014 | | Sponsor/Monitor's Acronym(s) |
| | | Sponsor/Monitor's Report Number(s) |
| Distribution/Availability Statement Approved for public release, distribution unlimited | | |
| Supplementary Notes See also ADM001433, Conference held International Conference on Methods of Aerophysical Research (11th) Held in Novosibirsk, Russia on 1-7 Jul 2002 | | |
| Abstract | | |
| Subject Terms | | |
| Report Classification unclassified | Classification of this page unclassified | |
| Classification of Abstract unclassified | Limitation of Abstract UU | |
| Number of Pages 5 | | |

that corresponded to the Reynolds' number on the wing chord $Re_c = 0.85 \cdot 10^6$; $0.99 \cdot 10^6$; $1.12 \cdot 10^6$; $1.17 \cdot 10^6$; $1.22 \cdot 10^6$; $1.25 \cdot 10^6$; $1.319 \cdot 10^6$. All tests were performed under conditions of natural boundary layer transition from laminar to turbulent on the model surface.

Investigation results

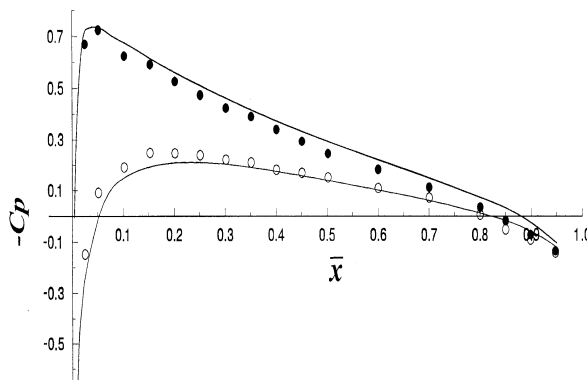
Distribution of surface pressure. The aim of the investigating opening stage was analysis of the boundary layer on the flowing wing under natural conditions and determining the angles of attack region in which flowing along the wing bears a continuous character. For this, the pressure distributions on the leeward and windward wing sides at the angles of attack 0° , 2° and 4° in the velocity range of the free-stream from 10 up to 50 m/s were measured and conditions under which a regime of continuous flowing along the wing realized were revealed.

To evaluate an influence level of the flow bounds and, as a consequence, to reveal a possibility of interference, special methodical experiments were performed and necessary corrections were obtained. Entering of these correction allows to argue that the obtained pressure distributions along the tube test section, including a model location region correspond to the endless flow creation.

The main task of the conducted investigations was analysis of possibility to describe properties of the considered flow by different theoretical methods and to evaluate their effectiveness. The comparison results of calculated (line) and experimental (symbols) values of the pressure coefficient along the wing chord at $\alpha = 2^\circ$ and at stream velocity of 35 m/s in the form of dependence $C_p = f(\bar{x})$ was shown in Fig. 1.

The calculated values were obtained in the frames of the compressed liquid model [1, 2]. On the whole, on a larger part of the wing profile a satisfactory agreement between two groups is observed. As it is seen from other results, some divergence typical for a tail model part and caused, as a rule, by a setting way of a flux downwards from the trailing wing edge is observed in our case too. Besides, in some cases the experimental data contain the error which oversteps the limits of occasional error of the C_p value measurement.

Figure 2 shows analogous dependences for the Mach number 0.7 and at zero angle of attack of the model. The experimental values of the pressure coefficients is marked with a dark circle, other symbols show the calculating results performed by analysis of different accounting methods of the flow compression [3–9], and also by the authors' numerical-analytical calculation. By this, the indicated calculating values were obtained using base (approximated) values C_p at nominal zero flow velocity. Satisfactory agreement of calculating distributions $C_p(x)$ with the experimental ones justify that on the most part of the profile surface all methods



describe a factor of the medium compression qualitatively right as the Mach number increases. Therefore, the obtained data differ from each other essentially.

At that, non-linear theories of the second (and third) approximation order (Karman – Tsien, Kuheman – Weber) have indisputable advantages in comparison with other simplified

Fig. 1

models modeled equally the pressure distributions on the largest wing surface. These theories can be considered as the most acceptable technique to account the compression factor, at least, for non-thick bodies of the wing profile type and small angles of attack. It should be also considered the numerical-analytical calculation that also shows a good agreement with the experiment. The same data were obtained at other studied M_∞ numbers.

Possibility to describe a pressure distribution character in the vicinity of the leading profile edge with the noted methods shall be considered too. As for directly spreading line, some non-linear theories (Karman – Tsien, Kuheman – Weber) and the numerical-analytical calculation with an acceptable accuracy reflect the pressure level even in this area. However, there is no assurance that these methods are reliable in the direct vicinity of the spreading line. Other analyzed theories do not reflect dynamics of the pressure coefficient growth on the spreading lines even qualitatively. In particular, according to the Leiton method the pressure level in this area is even decreased with M_∞ increasing that is obviously contrary to a physical sense.

Field of velocities and downwards in a model vicinity. The calculating and experimental data on the flow downward distribution in different transverse wing sections at minimum and maximum Mach numbers of the flow at which this value was measured are shown in the form of dependence $\varepsilon = f(\bar{x})$ in Fig. 3. Both qualitative and quantitative satisfactory agreement of calculating and experimental values of ε along the wing chord is seen. When the transversal coordinate y changes, a difference between them does not exceed nearly 0.5° . Therefore, a tendency of difference rise between a calculation and experiment observed at drawing near the model surface (y is decreased) is taken place. Insufficient accurate account in the calculating process of the boundary layer influence character explains the given fact. At the same time, taking into consideration a very complicated non-monotonous change character of dependence $\varepsilon = f(x)$, it can be reasoned that in the analyzed velocity range, the flux downward value is predicted by calculation quite satisfactory. This justifies in its turn, that the compression factor presented in the initial equations of gasdynamics, on the whole, describes correct a behaviour of the fluxes ownwards along the wing chords.

The following peculiarity observed at change of the flow velocity u_∞ shall be noted. Although change of value ε is of degree fraction in the most cases, almost systematical decrease (on absolute value) of the flux downward with increasing of U_∞ on the leading profile part ($\bar{x} < 0.3$) and its increasing on the other profile part ($\bar{x} > 0.3$) are observed.

Viscous effect. According to data [10] the effect of viscosity on the wing profiles is highly noticeable at the flux velocities of order 60 m/s and lower. However, to reveal a percent contribution of the viscosity value into distribution of the basic flow characteristics in the vicinity of the streamline body, separating it from the compression effect, can be made using

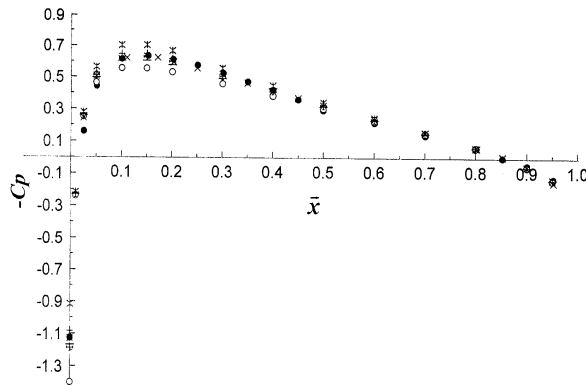


Fig. 2. Experiment (●) ; Calculation on the method: Prandtl – Glowert (○), Karman – Tsien (-), Spreiter (*), Weber (+), Leiton (*), on the model of compressed liquid (x).

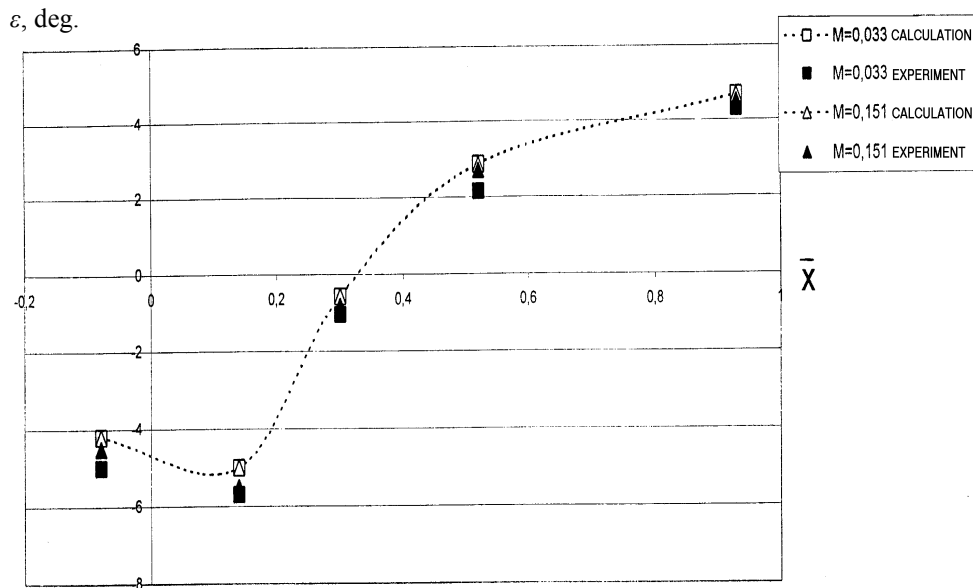


Fig. 3. Comparative data on the flux downwards distribution along the wing chord at $\alpha = 0^\circ$ and $y/c = 0.08$

special experiments in the wind tunnel of variable density. As for wind tunnel T-324, variation of the flow regime velocity leads to a change of the Reynolds' number, and consequently, to the viscosity change, because the characteristics of the boundary layer on the flowing surface are changed. Because there is not special devices, it would be useful to analyze the investigation results on one and the same model at different Reynolds' numbers, and by doing so to consider a contribution into the flowing process of the viscous value.

Figure 4 presents the data on distribution of the pressure coefficient C_p along the wing chord obtained at two numbers Re_c on the regime $M_\infty = 0.5$ and $\alpha = 0^\circ$.

Though a difference in Re_c numbers is not great, on the whole, a very good agreement of two groups of experimental data between each other is taken place. Thus, it can be argued, that, at least at moderate subsonic velocities, a change of the flow characteristics in the vicinity of the wing profile is due to the compression effect. It additionally confirms a truth of conclusion on modeling correctness of the medium compression process using theoretical methods, based

both on the model of ideal incompressible liquid with corresponding correctness and of gasdynamical equations.

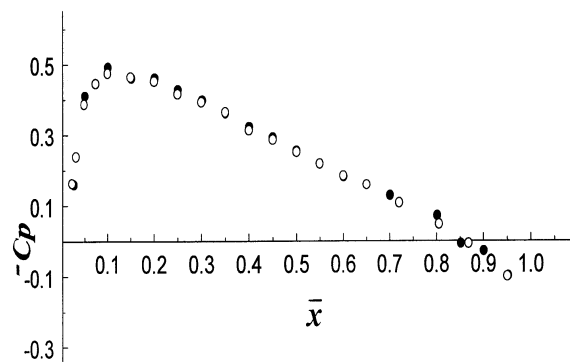


Fig. 4. Comparative data on the pressure coefficient. The Reynolds' numbers values: distribution along the wing chord.

Value of the Reynolds' number:

$Re_c = 0.99 \cdot 10^6$ (\bullet), the present work,

$Re_c = 5.00 \cdot 10^6$ (\circ), data [11].

Conclusion

In the flow velocities range from subsonic up to pre-critical, numerical-analytical and experimental investigations of flowing around the models of the symmetrical wing profile with revealing accent of the media compression role were made. Satisfactory agreement of the calculating results performed in the frames of the model of compressed liquid and of the experimental data indicates that in the range of relatively small velocities corresponding to the numbers $M_\infty \leq 0.15$, the compression factor presented in the initial equations, on the whole, describes correctly a basic flow parameters behavior (pressure coefficient, downward) on the most wing part. It was shown that for the bodies of a streamline contour which thickness does not exceed 12 %, there are no grounds to consider that a compression effect, at least up to the Mach numbers $M_\infty = 0.15$, is somewhat essential.

It was revealed the essential differences of the calculated data in the vicinity of the leading edge of the wing profile, including the own spreading line, that are caused by admission of a disturbance smallness, that is not true in a number of analyzed methods. Evidently, that this effect will be increased as the relative streamline body thickness increases. At the same time, some non-linear theories and numerical-analytical calculation reflect the pressure level with an acceptable accuracy even in this area. On the whole, taking into account the satisfactory accuracy of a result prediction in the vicinity of small inclinations of the flowing along the surface, it follows that the compression coefficient can not be accepted only in the function form from the Mach number for the whole area of investigated flow, as it is accepted in a series of the classical approaches.

Comparison of calculating results of the pressure distribution on the wing profile surface made on the basis of classical compression laws with the experimental data indicates that at a greater flow velocities, right up to $M_\infty \approx 0.8$, these laws based on using of the correction factor, mainly, model adequately the medium compression factor as the Mach number increases. Therefore, difference of the obtained data from each other is essential. Accuracy of the result prediction in the field of small inclinations of the flowing surface essentially depends on the approximation order of the theory itself, that, as a rule, is built on the disturbance smallness acceptance. Non-linear theory of the second (third) approximation order claimed to a higher accuracy, can be considered as the most acceptable technique to consider the compression factor, at least for unthick bodies of the wing profiles types. The numerical-analytical calculation which is in a good agreement with the experiment shall be also referred there.

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